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G. Hellard

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16. Abstract

The principles of aircraft element design using composite materials are outlined. Glass carbon and Kevlar composites are discussed. Sandwich and monolithic structures are described and application criteria are detailed. Drawings detailing the composite elements of the A-310 to A-330 and ATR-42 aircraft and the particular design of those elements are presented.

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THE DESIGN OF COMPOSITE STRUCTURES

by

G. Bellard

1. INTRODUCTION /1*

The use of composite materials has been becoming generalized in civilian aircraft for the past few years. A very extensive research effort has been and is continuing to be made by designers to acquire the knowledge needed for designing composite materials. I am going to introduce the main design rules.

The design of a composite piece involves selecting the material, the main construction principles and a related manufacturing procedure, and to take certain precautions for the drawing and the calculation.

2. SELECTING THE MATERIALS

2.1 Selecting the Fibers

The fiber is selected as a function of the nature of the piece: problem of strength, stiffness or strength and stiffness.

Glass or kevlar have a good strength, but have little stiffness. They will be selected for cockpit fittings type pieces, fairing or fuselage wing connecting fillet, for example. The latter type element should not be too rigid to fit wing and fuselage distortions without transferring stresses.

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*Numbers in the margin indicate pagination in the original text.

First generation carbon fibers are more resistant and are especially more rigid than glass or kevlar. They will be selected for pieces requiring stiffness and strength: for example, rudders, wing flaps, fin spar or horizontal stabilizer boxes.

Second generator fibers which appear increase the strength while maintaining the same stiffness, or increase both strength and stiffness. They will make it possible to make highly loaded primary structures, such as a wing box for example.

2.2 Selecting the Resin /2

The resin selection conditions the following main characteristics:

-Temperature strength of the laminate after aging,
-shock behavior,
-Smoke emission and fire resistance for inside cockpit fittings.

The two most commonly used resin families in civilian aviation are epoxy and phenolic resins.

The first generation epoxy resins polymerize at either 120°C or at 180°C.

The 120°C epoxy resins are self-adhesive on honeycomb. They have good mechanical properties after aging up to 80°C.

The 180°C epoxy resins generally require glue to stick to honeycomb. They have good mechanical properties after aging up to 120°C.

The second generation epoxy resins are used for new carbon fibers. They also provide better shock resistance.

For inside fittings, it is possible to use either an epoxy resin with a barrier for smoke emissions, or a phenolic resin.

2.3 Use of Preimpregnated Sheets or Fabrics

The materials used for manufacturing composites is in the form of unidirectional layers or fabrics preimpregnated with resins.

The unidirectional layers have more strength than the fabrics. For example, with a T.300 fiber impregnated with NARMCO 5208 resin, the unidirectional layer has a strength of about 20% higher than with a fabric woven with satin 5.

Moreover, draping is easier to optimize with layers. The stiffnesses compared between layers and fabrics are similar.

For manufacturing, on the other hand, draping of complex forms can be accomplished with the fabric. It is easier to pierce and to machine than layers.

It is easy to automate layer draping using a draping machine which is not the case with fabrics.

The use of fabrics is therefore systematic for pieces with complex forms (example: fairings, ribs) or for pieces whose strength is not essential.

For highly loaded pieces with a simple shape, layers or a mixture of layers and fabrics will be used.

2.4 - Use of Carbon/Kevlar Hybrid Coatings

By combining carbon layers and kevlar layers it is possible to obtain a material in between. The stiffness of the hybrid and its compression behavior are distinctly improved over that of Kevlar alone.

2.5 Examples of Using Various Composite Materials on an Airplane

-A.310-300 - (See Figure 1)

In Kevlar: fuselage wing fillet, engine pylon fairings and flap rails, radome, air intake for cockpit conditioning.

In Carbon: the airbrakes, rudders and elevator, gear doors, fin spar box.

In Hybrid: fin leading edge + engine pod.

-A.T.R. 42 - (See Fig. 2)

The A.T.R. 42, developed jointly by AEROSPATIALE and AERITALIA, has a large number of composite structures.

-In Kevlar: noses, flap fairings and commercial fittings (with phenolic resin), tail fuselage cone, fairing of main gear, radome.

In Carbon: the aileron, wing flaps, gear doors, rudders and elevators.

-In Glass: flooring.

3. SELECTION OF THE CONSTRUCTION PRINCIPLES /4

Three construction principles may be used:

-sandwich type structure,
-monolithic type structure,
-self-stiffened monolithic structure.

3.1 Sandwich Structures

A sandwich is made of two layers of composite materials glued to a honeycomb.

The honeycomb is usually Nomex to prevent corrosion (with carbon) and problems associated with lightning.

The layers consist of a stack of crossed layers. They are glued in a simultaneous polymerization operation to the honeycomb by means of the impregnation resin of the stratified layer, in the case of epoxy resins polymerizing at 120°C.

They are glued to the honeycomb by a glue film in other cases.

For very thin layers, a sealing sheet is added to the surface. An antistatic paint is provided for a glass and Kevlar structure.

A lightning protection using a metallic grid or an aluminum protection is achieved on the surface for pieces located in the lightning impact area.

Two manufacturing processes are used:

-Multiphase Process

The layers are polymerized then tested. The honeycomb is machined. Finally, the layers are glued with a glue film over the honeycomb in a second operation.

This type of process offers a good stratification of the layers. It is used for highly loaded structures.

-Monophase or Simultaneous Polymerization Process

The lower lining is draped. The honeycomb machined to form is then installed. Finally, the upper lining is draped over the honeycomb block. The unit is polymerized in a single operation.

The presence of honeycomb limits the pressure applicable to the linings during the polymerization operation and can create flaws in the stratified layer. Moreover, a "telegraphing" effect (lining undulations) is produced which makes the mechanical properties drop by about 15%.

The simultaneous polymerization process makes it easier to solve the problems of providing patterns on panel edges or on door frames in a panel (see figure 3). In this case, the inside lining comes back in contact with the outside lining by means of a slope over the honeycomb.

The panels are bordered by Z patterns reported in the case of a multiphase process.

For an airbrake box, for example, an optimum drawing can be achieved with only two linings by simultaneous polymerization (see figure 3).

For the same type of box, the multiphase process necessitates a spar and a ledge in addition to two linings.

The advantages of sandwich structures are numerous. The honeycomb provides a good compression behavior by stabilizing the linings.

It is easy to obtain a good stiffness on a panel by increasing the honeycomb height, making it thus possible to eliminate the ribs.

The drawbacks of sandwich structures appear for very low or very high stresses and they affect the honeycomb dimensioning.

The latter is dimensioned according to two criteria:

1) Shearing stresses in the case of flight,
2) Autoclave pressure required for polymerization of the linings while the piece is being manufactured.

For low stresses, criterion no. 2 is actually dimensioning and optimization: honeycomb mass versus lining mass is no longer optimum, particularly when the honeycomb passes through a thick structure.

For very high stresses, it is possible to increase the lining thicknesses, but runs the risk of skin/honeycomb separation. It is therefore more interesting to use structures self-stiffened by stiffeners.

A few typical examples of sandwich structures are given in figure 4 for AIRBUS:

-Kevlar engine pylon fairing,
-carbon rudder, made by M.B.B. (figure 5) for the A.T.R. 42
-kevlar leading edge,
-wing flap.

3.2 - Monolithic Structures

This structural design does not use honeycomb.

Monolithic structures are divided into two families:

-Monolithic structures with nonstiffened lining, supported by a multi-spar or multi-rib substructure,

3.2.1 - Multi-Rib Box type Monolithic Structure

A typical example is the inside air brake of the A.310 (see figure 6).

Two manufacturing processes may be used:

-Multiphase and monophase process

The advantages of monolithic structures are:

-Lightness for low stresses and low height between ribs.

-Manufacturing cost less than that of sandwich structures.

-Easy repairs using polymerized carbon plates fastened to the box by means of blind attachments.

The drawback of this type of structure is, for high stresses in the linings, to have to reduce the rib pitch. It is therefore necessary to use a structure with self-stiffened panels.

3.2.2 - Self-Stiffened Monolithic Structures

They are made of self-stiffened panels supported by two spars and two ribs. This type of structure is resistant to very high stresses. It is the typical design of a fin, tail unit or wing box.

Three types of panel stiffening are used:

-Solution I, solution I, solution .

-The skin thickness and draping vary as a function of the stresses. The height of the stiffener is also variable.

In the three cases, the stiffener is polymerized at the same time as the panel, which makes it necessary to produce sophisticated tools.

An example of this type of structure is the V.10 F wing (see figure 8).

The "FALCON 10 wing) is an experimental program carried out jointly (AEROSPATIALE/DASSAULT) and financed by the French government.

It consists of manufacturing a carbon wing which is expected to fly in May 1985.

The panels are U-shaped, assembled on 16 ribs and two carbon spars.

3.3 - Comparative Study of Various Types of Structures

I will give three typical examples:

3.3.1 - Air Brake A.300/600 - (See figure 9)

Before the serial production of inside A.300/600 carbon air brakes, AEROSPATIALE studied, manufactured, tested and flew several types of structures.

-Sandwich solution with unidirectional layers,
-Monolithic solution with layers and fabrics.

A comparison between the two layer solutions gives:

-The weight gains over metal are similar: 25% for the monolithic solution, 21% for the sandwich solution,

-The strengths are similar: 3 x Limiting Load for the monolithic solution, 4 x Limiting Load for the sandwich.

-The manufacturing costs are more advantageous for the monolithic solution.

The selection is therefore the same for this type of structure. The monolithic solution was adopted.

3.3.2 - AIR.42 Carbon Aileron (See fig. 10) /8

The AIR.42 aileron solution was selected after studying three different structures, each using unidirectional carbon layers.

The metallic reference structure is a box with 2 spars and 18 ribs, with smooth light alloy lining.

Carbon Solution 1

The box is a solid honeycomb sandwich The nose is a sandwich panel type, inserted by mechanical attachments.

Carbon Solution 2

The aileron is made of two sandwich panels interconnected at the nose level and fastened to a spar and a ledge by mechanical attachments.

Carbon 3 Solution

The aileron is made of a multi-rib monolithic box, sealed by a spar and upon which a nose is inserted by mechanical attachments.

The weight gains over metal are: 105 for solution 1, ;9.6% for solution 2 and 18% for solution 3.

The monolithic solution is therefore 9% lighter than the solid honeycomb sandwich solution, which is explained by the low stresses and thickness of the large section (130 mm approximately) which gives a large mass.

The sandwich panel solution is also at a disadvantage. In effect, the two carbon skins of each panel are excessive for the stresses to to hold.

From the price standpoint, the monolithic solution is approximately less costly than the sandwich solutions.

The monolithic solution was therefore retained for the airplane.

3.3.3 - Wing Flap (See figure 11)

A study, which did not result in the production of an element,

was conducted on an AIRBUS wing flap.

Three carbon layer solutions were compared to the existing solution. In all solutions, the TAB is made of a carbon skin and a solid Nomex honeycomb sandwich. The nose is also common to all three solutions.

Solution 1

The box is a solid honeycomb sandwich limited by two spars, two sealed ribs and 4 intermediate ribs.

Solution 2

The box is made of two sandwich panels fastened by screws to two spars and to 11 ribs.

Solution 3

The box is made of two Ω self-stiffened panels and fastened by screws to two spars and to 27 ribs like the metallic solution.

The weight gains over metal are: 8.8% for the solid sandwich, 16.3% for the sandwich panel solution and 20.8% for the self-stiffened Ω panel solution.

The results of this study were used for selecting the construction principles of AIR.42 wing flaps. They are made up of 2 carbon/honeycomb sandwich panels bolted to a spar and to a ledge.

3.3.4 - Conclusion

For each application case and as a function of the stress geometry, it corresponds to an ideal type of structure.

4. - SPECIAL DESIGN PROBLEMS FOR COMPOSITES

4.1 - Stacked Layer Design

The unidirectional layers are 20 times stronger in the fiber direction than in the direction perpendicular to the fibers.

It is therefore possible to layout the fibers in the stress direction and to thus create the ideal material adapted to a special strength or rigidity problem.

Figure 13 shows how the optimization was achieved in the three following cases based on 4 normalized directions 0, +45, -45, 90:

For the inside air brake, the normal stresses in the 2 perpendicular directions and the shearing require the same number of folds in all directions (quasi isotropic draping).

The stresses of a hinged rod are allows directed along its large /10 axis. It is therefore possible to put 80% of the fibers in this direction. The optimization is maximal and the weight of the carbon rod half of that of the light alloy rod it replaces.

The upper surface of the FALCON 10 wing has 15 to 74 layers. It is dimensioned for twisting stiffness, at the wing tip. The percentage folds at ± 45° is high: about 30%, and that at 0° is slow (26%). The wing root is dimensioned for strength. The percentage of folds at 0° is high: 48%.

4.2 - The Assemblies

Glue bonding is the most commonly used method as shown in the examples.

The introduction of concentrated stresses associated with the presence of iron fittings, however, requires a bolted connection.

The presence of holes in a metallic or composite material creates an overstress along the edge of the hole of about 3 times the running stress (see figure 12). Near the rupture, the metal passes into a plastic range, the overstress peaks and delays the rupture. For the composite, overstress spreads little and the rupture occurs earlier.

The decline in strength of a composite with an attachment hole with a running diameter (5 mm) is about 40% compared to the strength without a hole. It is therefore necessary to dimension the composite assemblies as a function of the presence of holes. Zones without holes become excessive.

Thus, a structure assembled by bolts with a diameter of 5 mm can undergo accidental damage, creating a hole with a diameter of 5 mm without losing strength, and accidental damage creating a hole with a 40 mm diameter, with a decline of only 20%.

4.3 - Carbon/Metal Assemblies

Metal, an isotropic material, remains suitable for making fittings of complex forms.

Carbon/light alloy assemblies are therefore common.

In this case, the following precautions should be taken:

-Aluminum and carbon in the presence of humidity create a galvanic torque which corrodes metal quickly. Direct contact between carbon and light alloys should therefore be avoided.

Protections provided:

.anode oxidation of the light alloy,
.painting of assembled elements,
.cementing mastic,
.if necessary: glass on carbon fabric.

-It is not possible to assemble attachments with interference

in the carbon, nor attachment which swell while they are being installed, such as rivets.

Since the assembly is pierced in a single operation, there is clearance between the bolt shaft and the hole, in the carbon, and the light alloy. This clearance gives the metal poor fatigue strength.

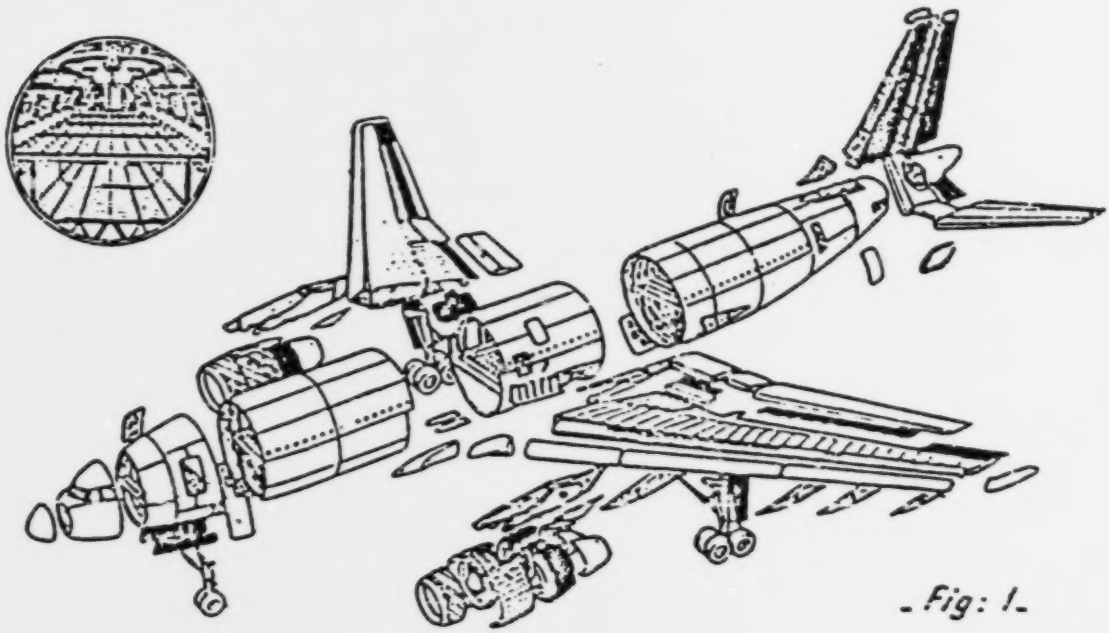
-The carbon expands little. The light alloy has a high expansion factor. Since the assembly is produced at 20°C, in flight at -50°C, the difference in the expansion factor creates for each flight a tensile stress in the metal which is added to the mechanical stresses. The iron fitting must therefore be dimensioned for for low stress fatigue. Conversely, carbon is dimensioned for rupture stresses.

-The design rules for the distance of an attachment relative to the free edge of the piece are different in metallic composites: 3 times the diameter for composite and 2 times the diameter for metal.

OF POOR QUALITY

COMPOSITE STRUCTURES

A 310_300 -



_ Fig: 1_

CARBON FIBRE
HYBRID CARBON / ARAMID
ARAMID
FIBREGLASS

ATR_42 -

_ Fig: 2_

COMPOSITES DESIGN - VARIOUS TYPES OF SANDWICH STRUCTURES

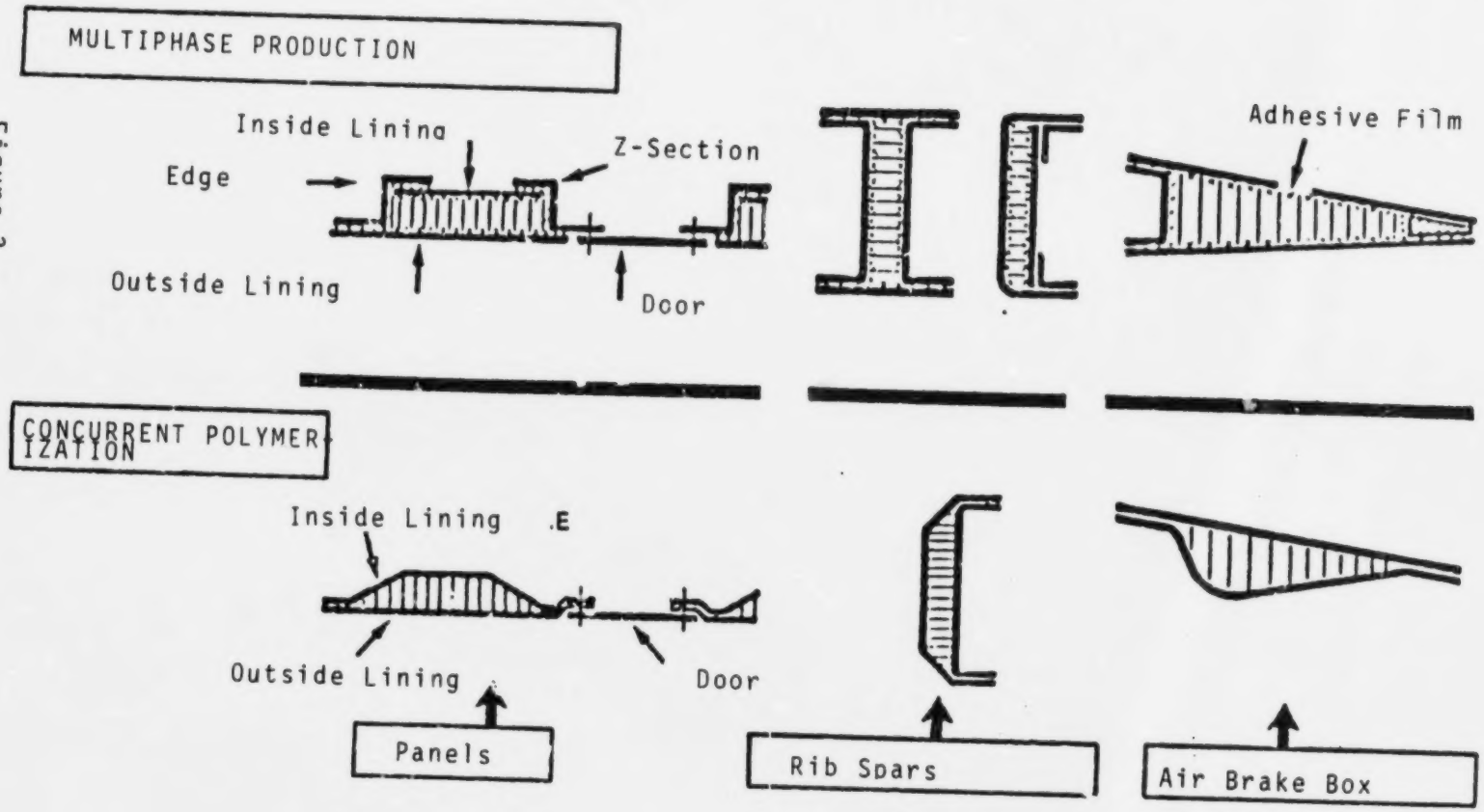


Figure 3

COMPOSITE DESIGN
SANDWICH STRUCTURES

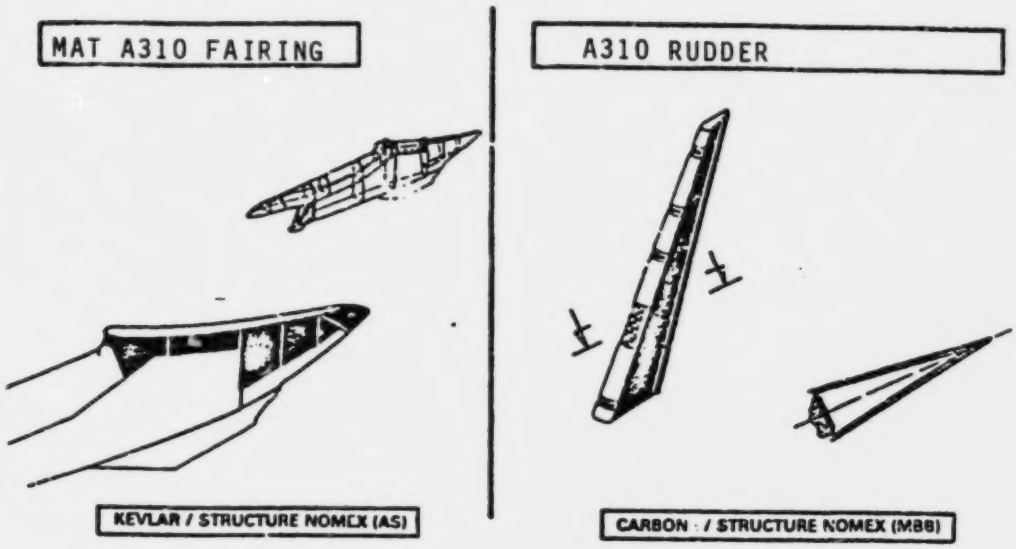


Figure 4

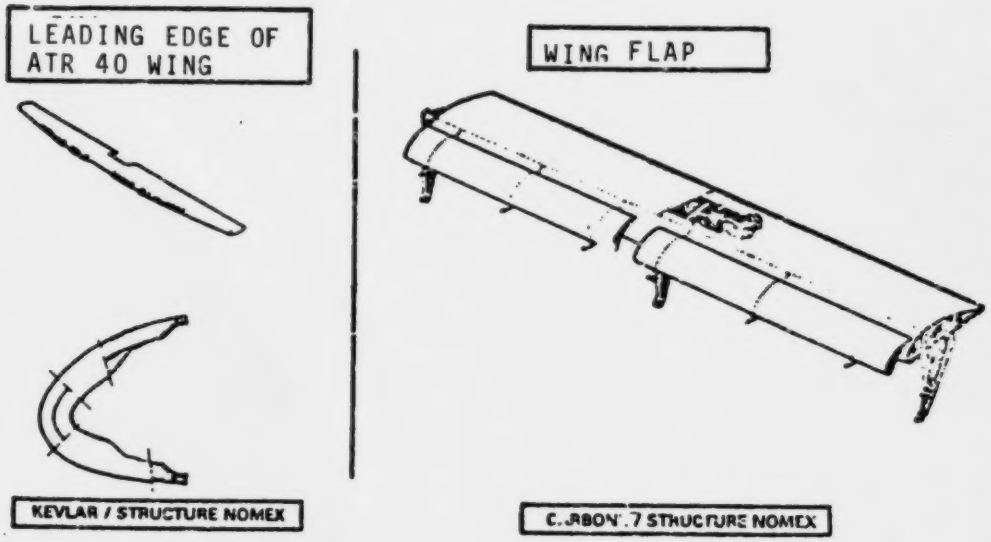


Figure 5

COMPOSITE DESIGN
MONOLITHIC STRUCTURES

- A.310 Inside Airbrake

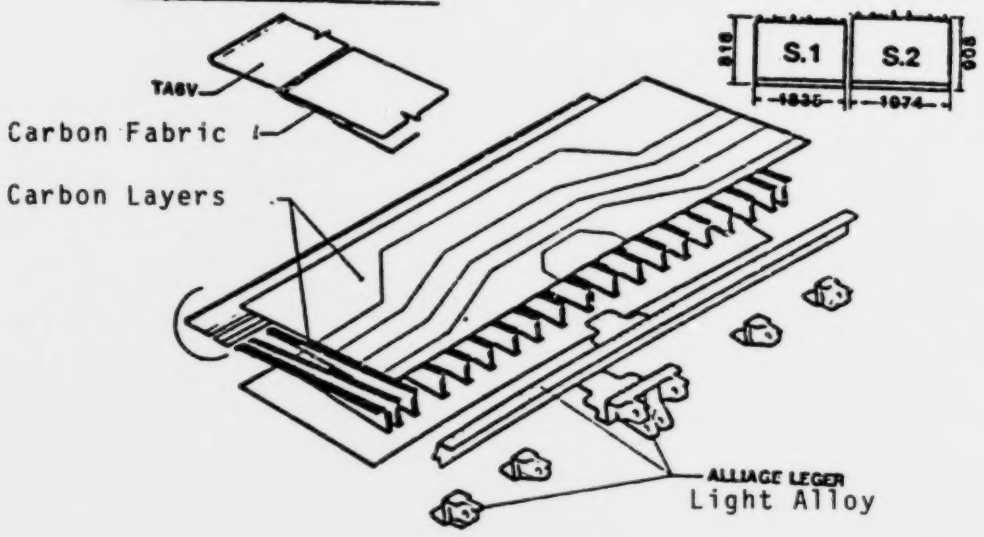


Figure 6

SELF-STIFFENED MONOLITHIC
STRUCTURES

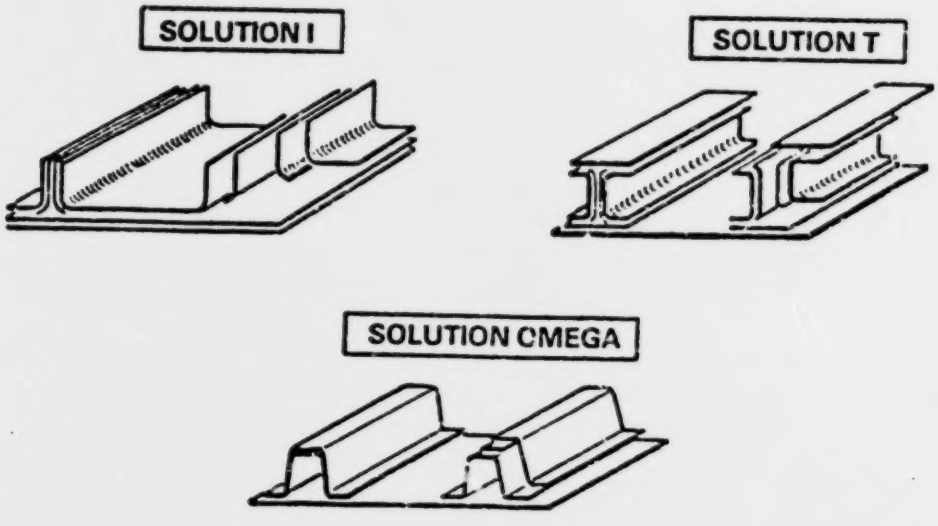
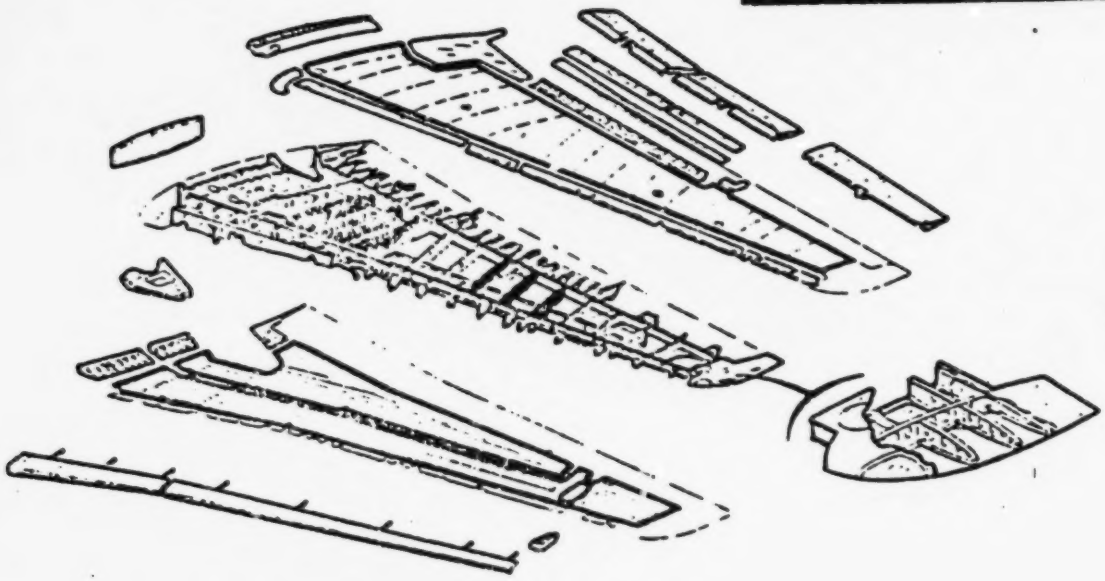


Figure 7

COMPOSITE DESIGN

CARBON WING FOR FALCON 10
EXPERIMENTAL PROGRAM
AMD/BA COOPERATION
AND FINANCED BY THE FRENCH GOVERNMENT



CARBONE

LIGHT ALLOY

Figure 8

COMPOSITE DESIGN
SELECTING THE STRUCTURE
A300/600 INSIDE AIRBRAKE

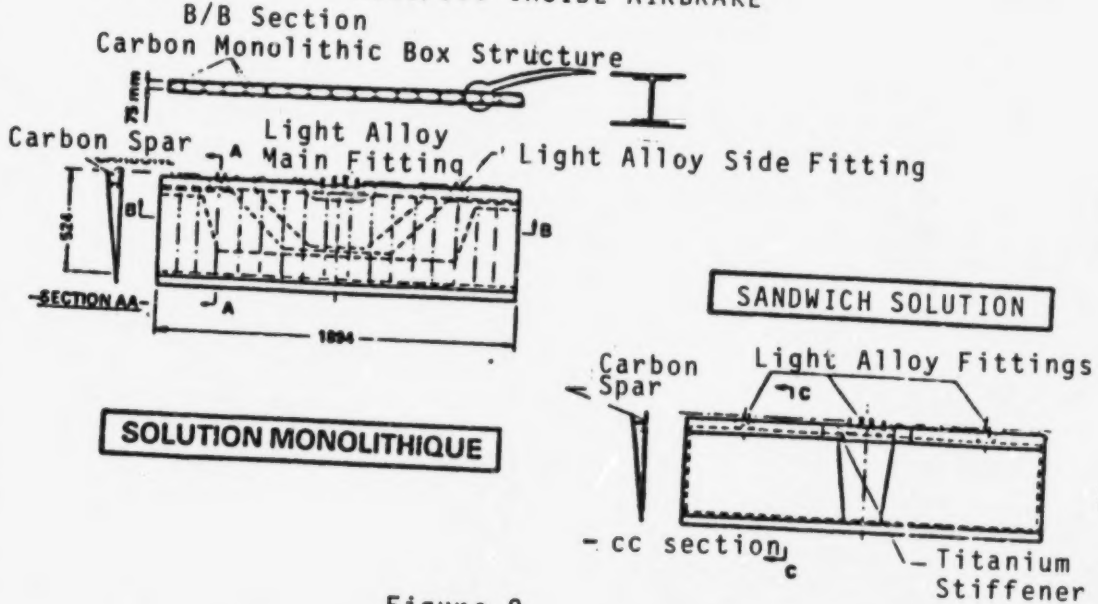


Figure 9

SELECTING THE STRUCTURE - ATR42 AILERON

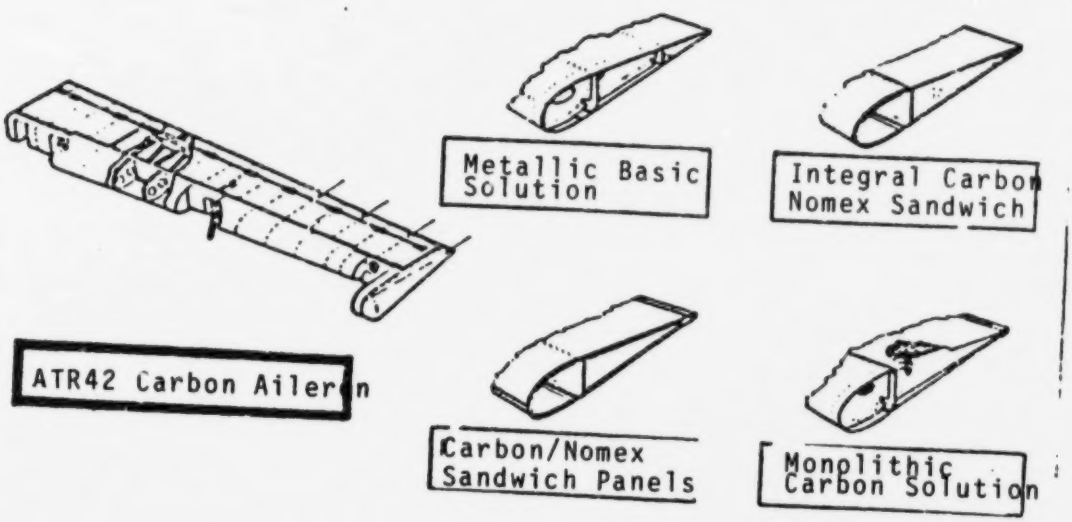
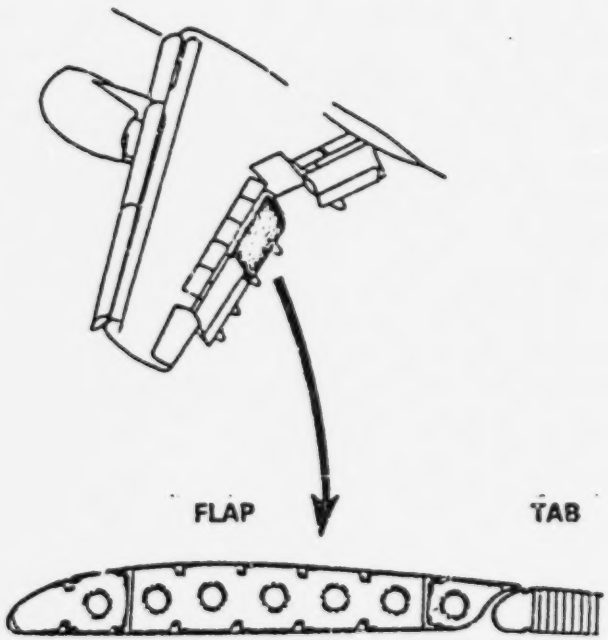


Figure 10

COMPOSITE DESIGN

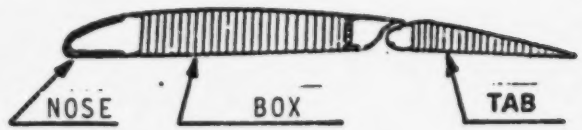
SELECTION OF WING FLAP STRUCTURE



METALLIC SOLUTIONS

SOL. 1 : SANDWICH

SOLID NIDA



SOL. 2 : SANDWICH PANELS

SOL. 3 : MONOLITHIC PANELS STIFFENED BY

CARBON SOLUTION

Figure 11

OF POOR QUALITY

COMPOSITE DESIGN

OPTIMIZING THE LININGS

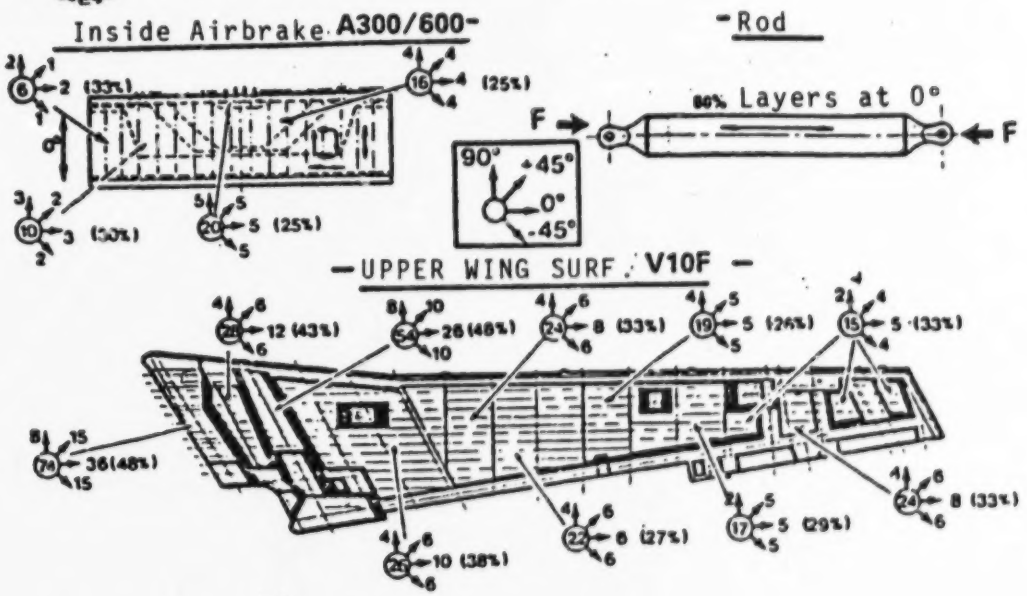


Figure 12

ASSEMBLIES

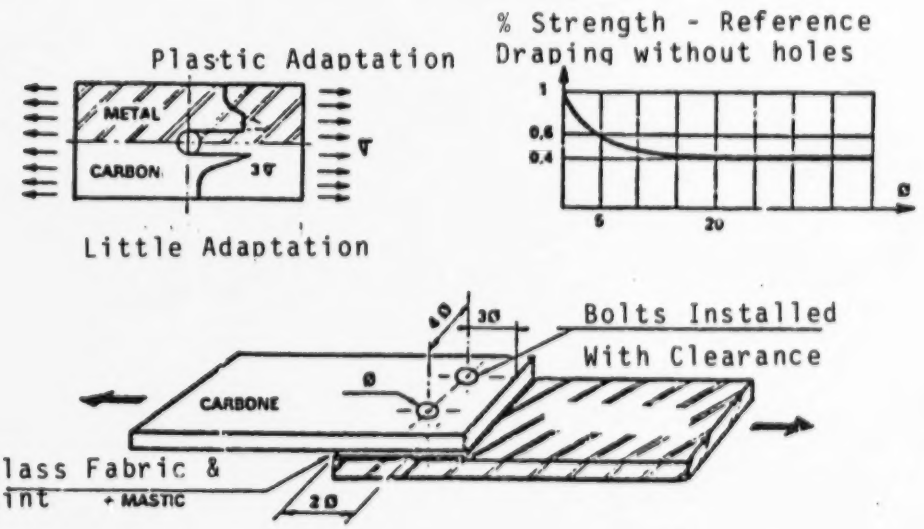


Figure 13